A NOVEL APPROACH TO ROTORCRAFT DAMAGE TOLERANCE

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Abstract

Damage-tolerance methodology is positioned to replace safe-life methodologies for designing rotorcraft structures. The argument for implementing a damage-tolerance method comes from the fundamental fact that rotorcraft structures typically fail by fatigue cracking. Therefore, if technology permits prediction of fatigue-crack growth in structures, a damage-tolerance method should deliver the most accurate prediction of component life. Implementing damage-tolerance (DT) into high-cycle-fatigue (HCF) components will require a shift from traditional DT methods that rely on detecting an initial flaw with nondestructive inspection (NDI) methods. The rapid accumulation of cycles in a HCF component will result in a design based on a traditional DT method that is either impractical because of frequent inspections, or because the design will be too heavy to operate efficiently. Furthermore, once a HCF component develops a detectable propagating crack, the remaining fatigue life is short, sometimes less than one flight hour, which does not leave sufficient time for inspection. Therefore, designing a HCF component will require basing the life analysis on an initial flaw that is undetectable with current NDI technology.

Introduction

There are several methods being postulated to safely and economically design and maintain the fatigue life of rotorcraft. These methods are known as stress-life, flaw-tolerance, damage-tolerance, and fault-tolerance methods. Traditionally, the fatigue behaviour of rotorcraft has been designed and managed using stress-life methods. Stress-life methods directly relate service loads to a safe operating life that is based on a linear representation of cumulative damage [a, b]. In this method, laboratory specimens are cycled under constant-amplitude loading until failure occurs. The resulting data is then summarized as a plot of stress versus cycles, which is commonly known as an S-N curve. Based on this laboratory data, the anticipated service loads are used to predict the time to failure for an actual structure. In particular, each loading cycle in the service spectrum is presumed to cause a specific amount of damage, based on the laboratory-defined S-N curve, which is linearly summed to determine when the structure fails. Using this method, a specific time to failure is determined and then a factor of safety is applied by retiring the structure when a specific number of cycles have been reached.

The flaw-tolerance method attempts to add additional conservatism to the stress-life method by using an S-N curve that is based on using flawed laboratory specimens [c]. The flawed specimens are presumed to be representative of the usage environment of the actual structure, and are used to define the fatigue life of a damaged component. Based on the fatigue life of the flawed specimens, inspection intervals can be determined to retire or repair the component,

based on detected damage. Therefore, the part can be designed with a less conservative S-N curve because field inspections will be performed, in contrast to the traditional stress-life method which has no inspection requirement.

The damage-tolerance method is a fracture-mechanics-based design philosophy that is founded on the ability of a structure with a crack to maintain integrity. This method for design and fatigue-life management has been promoted extensively by the United States Air Force [d]. The assumed crack size inherent in the structure is defined by the detection capability of non-destructive inspection (NDI) methods. The Air Force requirements assume an initial crack size for fuselage and wing structures to be 1.27 mm, based on component accessibility and NDI capabilities. For rotorcraft structural components, a proposed NDI crack size of 0.4 mm is currently recommended that is based on manufacturers' experience [e]. Utilizing these NDI initial flaw sizes, a fracture-mechanics approach is used to determine the life of the component, and inspection intervals are defined that are based on component accessibility and cost.

In a high-cycle fatigue environment, such as that experienced by a rotorcraft spindle lug, the damage-tolerance method described previously becomes impractical [f] because the number of cycles generated per hour of operation can exceed 60,000 [g, h], which results in a predicted component failure within hours. This life estimate is overly conservative, since many high-cycle fatigue components such as propellers and helicopter rotor hubs are designed with safe-life methods and are in service safely for thousands of hours before retirement. Therefore, to implement a damage-tolerance design method for these high-cycle fatigue environments, other means of determining initial crack sizes and inspection intervals must be investigated.

The fault-tolerance method is a new method to damage-tolerance design that is intended to ease the burden of frequent inspection of high-cycle-fatigue structures. It is based on the concept of an equivalent initial flaw size (EIFS) [i], where the damage is presumed to be equivalent to a crack-like defect. Similar to the flaw-tolerance method, flawed specimens are used in a damage-tolerance analysis to predict a time to failure for different damage conditions and to define inspection requirements. This method differs from traditional damage-tolerance methods in that the inspections are defined for damage such as corrosion and foreign-object damage, which are precursors to cracks, and not specifically for cracks.

In this paper, stress-life, flaw-tolerance, damage-tolerance, and fault-tolerance methods for fatigue-life design and life maintenance of rotorcraft structures that are subjected to a high-cycle loading environment are examined and evaluated. In particular, evaluations of the four methods are conducted for a fictitious rotorcraft spindle lug in which relative cost, safety and practicality of the four methods are compared. In addition, the impact that an altered-usage environment has on each of the design and life management methods is examined. To accomplish these goals, details of design and life-management methods that are based on the four methods are discussed first. Then, the classification of flaws that are used to define fatigue life and inspection intervals is presented. Next, the four methods are applied to the rotorcraft spindle lug, the results are compared, and the spindle-lug economics are discussed. Finally, a general discussion of the overall cost and safety of each method is presented.

Design and Life-Management Methods

There currently exist two methods for designing rotorcraft structure: stress-life methods, also called the safe-life methods, and damage-tolerance methods. The stress-life methods are based on the empirical relationship between stress and failure that is defined by testing of laboratory

specimens and by full-scale structural testing. The damage-tolerance methods relate stress and failure via fracture mechanics, fatigue-crack growth, or fracture. Within each of these classes of methods exist several variances, however, the fundamental method used in each class is the same. One important variance of stress-life methods that is considered in the present study is known as flaw-tolerance methods. Likewise, one important variance of damage tolerance methods that is proposed in the present study for use in design and life management is referred to herein as the fault-tolerance method. The fundamental aspects of each of these methods are discussed subsequently.

Stress-life methods

Palmgren [a] and Miner [b] developed a method that relates the failure of laboratory specimens under constant-amplitude loading to the failure of structures. This procedure allows a designer to relate the stresses obtained from a design model directly to the number of cycles to component failure. For instance, if a component which experiences only one constant-amplitude-load value is required to last 1,000,000 cycles, Figure 1 indicates that the vibratory stresses may not exceed 90 MPa (see the S-N as-manufactured curve in Figure 1). A more complicated structure would require a safe-life analysis where the contribution of individual loads must be accounted for in the analysis [j].

The management of stress-life-designed components is also straightforward. The design analysis yields a time to failure that is based on the S-N curve and component usage, with some factor of safety applied. This time to failure is designated as a retirement time at which the operator removes the component from service and replaces it with a new part. This component replacement is repeated for the life of the structure. Periodic visual inspection of the part, if accessible, is usually advised but is not required.
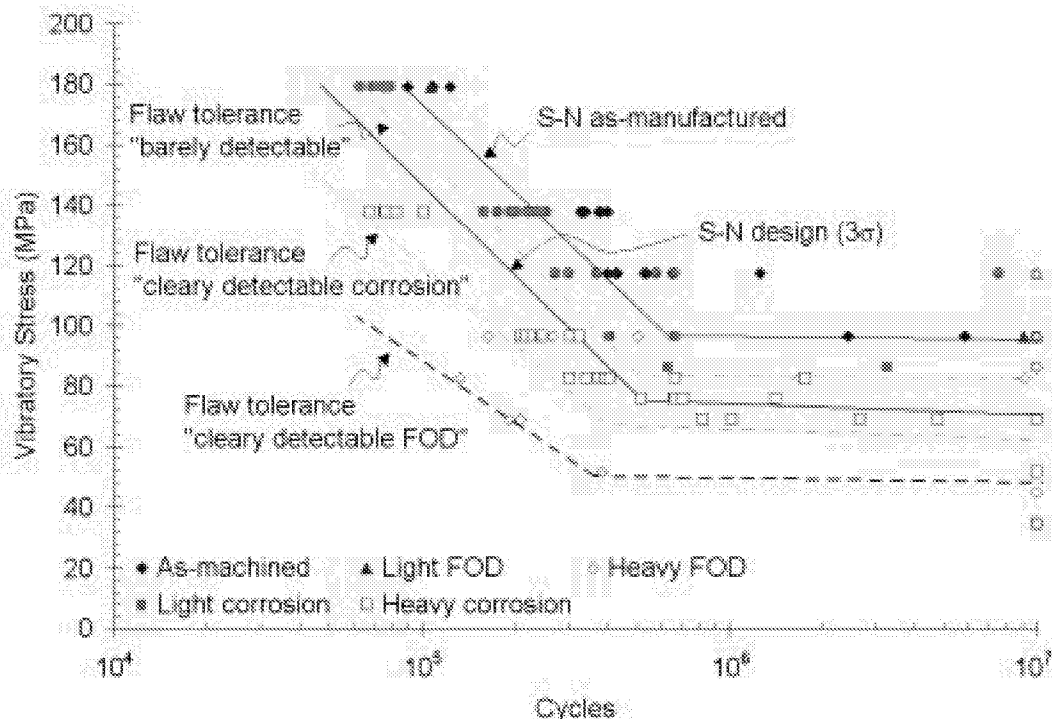
Flaw-tolerance methods

To design a component via the flaw-tolerance method, generation of stress-life data (S-N curves) for the as-manufactured condition and barely detectable and clearly detectable field damage must be generated. Barely and clearly detectable damages are defined by the probability of detection for each type of damage by using visual inspection. For this case, data generated for an aerospace-grade aluminum has been obtained and is plotted in Figure 1 for the as-manufactured condition, light corrosion, foreign-object damage (FOD), heavy corrosion and FOD to quantify the barely and clearly detectable damage. Component retirement times are determined by using the linear cumulative-damage approach, as defined under stress-life methods, based on the barely detectable damage curve, light corrosion and FOD.

Flaw-tolerance methods are also based on the notion that components can be designed safely by using a simple stress-life method. However, instead of using the standard stress-life factors of safety, which can be overly conservative, the component is managed by inspection for "flaws," such as corrosion pits or foreign-object damage and subsequently retired from service [c]. This method of structural design and management is based solely on the presumption that flawed laboratory specimens behave in a similar fashion to a flawed component that is in service. This presumption will be further discussed in the subsequent section.

Life management of the flaw-tolerance-designed component must then be governed by periodic inspection. Inspection intervals are based on a linear cumulative-damage analysis of the structure by using the clearly detectable flaw-tolerance data (Fig. 1). The failure time is used to define an inspection interval. Generally, three inspections must occur prior to the predicted

failure time. Furthermore, because the amount of life remaining in a component with a specific damage state is known from the S-N testing performed; a "retirement-for-cause" program may be implemented while monitoring damage evolution. Retirement for cause describes the process of removing a component from service because of clearly detectable damage that is beyond repair. For instance, by using the simple example from the stress-life section presented herein (90 MPa applied stress), if light corrosion is observed during an inspection at 250,000 cycles, the part is not necessarily retired because it has a usable fatigue life of 300,000 cycles remaining (Fig. 1). However, the inspection interval of the component would then be decreased to 75,000 cycles, based on the requirement for three inspections during the remaining safe-life estimate of the damaged structure.


Figure 1: Flaw-tolerance Stress-Life data for an aluminum alloy.

Damage-tolerance methods

Damage-tolerance methods for design and life management define a crack-growth life as the service time required for an initial-crack size, that is readily detectable by nondestructive inspection methods, to grow to failure. The fatigue life is computed by using fatigue crack-growth-rate data [k] (as shown in Figure 2 this data relates the crack growth rate, da/dN to the stress intensity factor, ΔK, that is a function of geometry, stress and crack size), the component geometry, and the usage environment as input into a commercially available computer code such as NASGRO [l]. NASGRO was used for the subsequent analyses presented herein. The life of the component is managed via inspection intervals that are set as a minimum of three opportunities to detect the crack during the predicted life [d].   For rotorcraft structural components a proposed NDI-based crack size of 0.4 mm is currently recommended, based on a manufacturer's experience [e]. It is presumed this value is based on a risk assessment of the NDI

methods used in service and the manufacturing tolerances used in production. Unlike for stress-life methods, the component is not necessarily retired at the end of the design life, but is managed via increased inspections until the economics of inspection exceed that of component replacement. The component is retired or repaired when a crack is found.
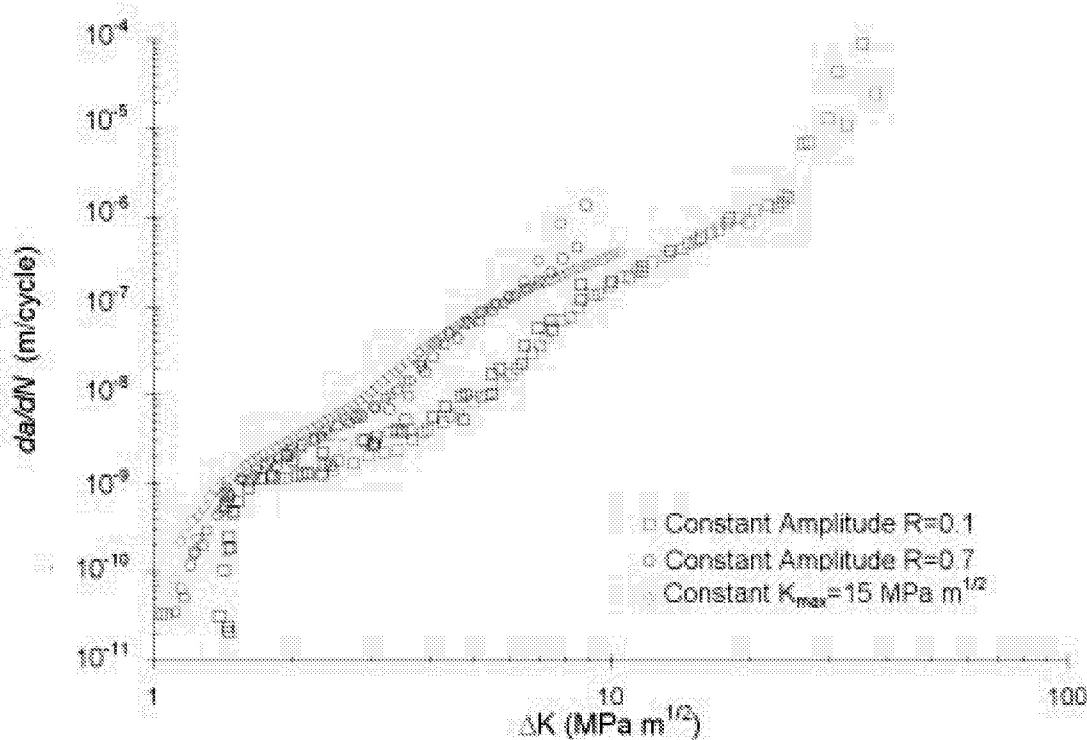
**Figure 2:** Fatigue crack-growth rate data for an aerospace grade aluminum alloy.

Fault-tolerance methods

Computer scientists have defined fault tolerance as "the ability of a system to respond gracefully to an unexpected hardware or software failure" [m]. Herein, this term is used for an aerospace application to describe a merging of the flaw-tolerance and damage-tolerance methodologies [n] to reduce the burden of inspection for "undetectable" cracks. The focus of the fault-tolerance method is to design a structure that is based on manufacturing tolerances and to manage the part based on service conditions. However, the inspection for cracks that are detectable at manufacture is unrealistic in the field. Typically, a producer can detect damage on the order of 0.1 mm in depth, with a 90% probability, whereas a field inspection may miss a flaw as large as 1.0 mm.

The equivalent initial-flaw-size (EIFS) concept is founded on the assumption that damage rapidly evolves into crack-like defects that propagate under fatigue loading [i]. Therefore, all stress-life data can be used to predict the size of a crack that must have been present in the specimen to cause a failure. In other words, the EIFS is the crack size needed to predict the failure of a fatigue specimen. This method can also be applied to flaw-tolerance specimens, where pre-existing damage caused the specimen to fail prior to the as-manufactured condition. Utilizing the NASGRO [l] computer code, the computation of an EIFS value for each failed S-N specimen (Fig. 1) was undertaken and the results are plotted in Figure 3 as a cumulative

distribution function. It has been postulated that the stress level has a significant effect on the computed EIFS value [o]. For example, the EIFS value is not representative of the physical condition of the specimen but of the loading applied to the specimen. If accurate fatigue-crack growth-rate data [k], as shown in Figure 2, is used for the analyses there is little dependence on stress evident for each condition, with the exception of the heavy FOD, as illustrated in Figure 4. The heavy FOD specimens may exhibit residual-stress effects and exhibit an order of magnitude or more scatter in both the S-N and EIFS plots. The usage of this data is discussed later in the paper.

The EIFS for the as-manufactured condition is then used to predict a durability life for the structure; e.g., assuming an inherent defect in the component will propagate to failure. This durability life is used during design to size the component. The EIFS values for the states of damage, barely and clearly detectable, are then used to predict a time to failure for each of these conditions and to set inspection intervals for this damage accordingly. The inspection for damage does not preclude the inspection for cracks, of a detectable size, and when either is detected the part is repaired or scrapped. One reason for using this method, in lieu of a classic damage-tolerance method, is that high-cycle fatigue components may experience usage environments where crack-based inspection is unrealistic. For example, the crack size required to achieve an economically viable service life is undetectable by current technologies.
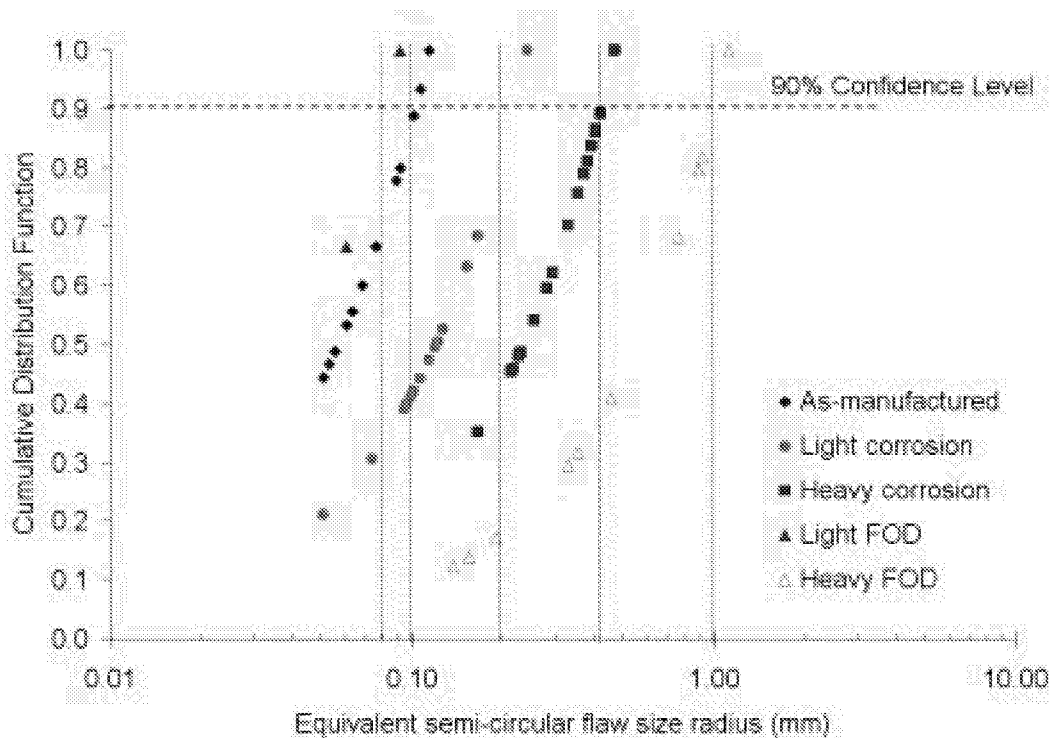
Figure 3: Equivalent initial flaw size computed from Stress-Life data (Fig. 1).
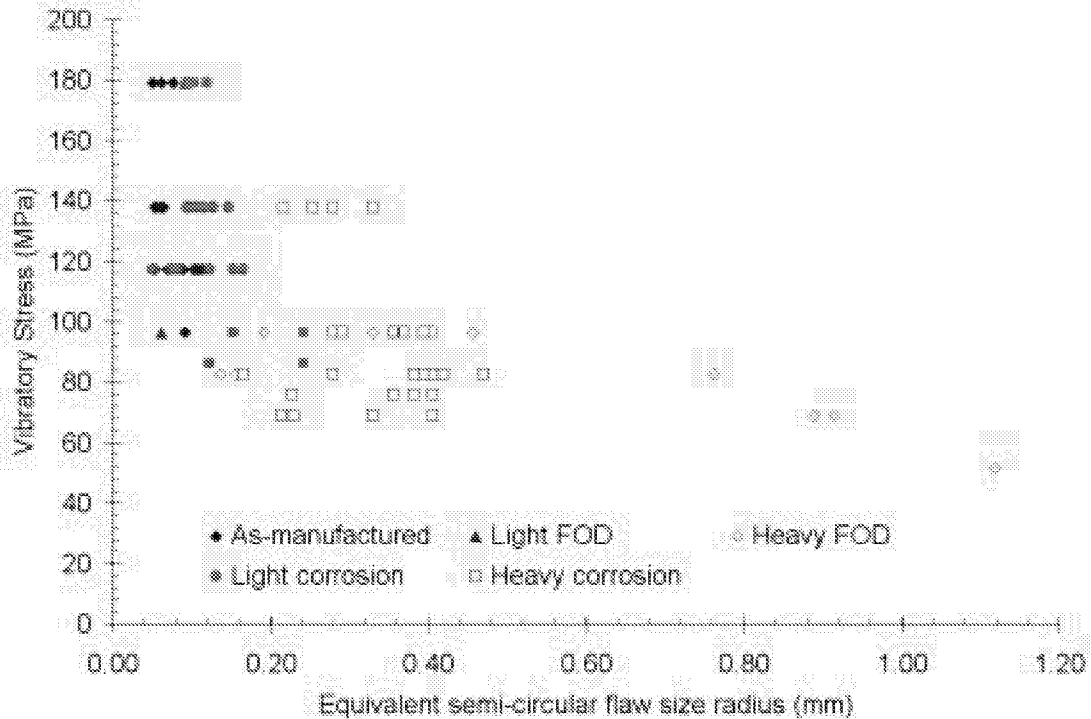
Figure 4: Vibratory stress versus equivalent initial flaw size.

Classification of Flaws

The flaw- and fault-tolerance methods assume that damage introduced at the specimen level is representative of that seen in service. Further, it is assumed that this laboratory damage must conservatively encompass all contingencies within reason. Corrosion and foreign-object damage are the typical forms of damage for rotorcraft that lead to fatigue cracking and subsequent failure. A survey of components that have been removed from service by a rotorcraft manufacturer illustrates this trend, as shown in Figure 5. For most cases, the damage found did not exceed one millimeter in depth, either corrosion or FOD. Therefore, the classification of light and heavy laboratory damage was 0.13 mm and 1.02 mm respectively. While this estimate of "flaws" will not encompass every possible scenario, most structures are not designed with every contingency in mind.

The light corrosion found in the field was typically widespread in areas where paint had been removed or in an environment where salt spray is evident. In this instance, there is documented field data to compare to the laboratory damage, and photographs of both the actual damage and that generated in the laboratory are shown in Figure 6. Heavy corrosion was mostly restricted to exposed structure subjected to heavy usage, such as a military application. The foreign-object damage was mostly due to installation and/or maintenance of nearby components, resulting in fine scratches or gouges in the surface. Heavy FOD was seen in cases where impact of the structure was noted (a bird strike) or poor maintenance practices were observed. Foreign-object damage was introduced into specimens with a CNC machine in an attempt to avoid the residual stresses introduced via dynamic impact. However, there is significant variation in inducing foreign-object damage in the laboratory [p] as is seen in the scatter of the S-N data (Fig. 1) and

corresponding EIFS values (Figs. 3 and 4). The corrosion pits were introduced into specimens with electrochemical etching. Contrary to foreign-object damage, this method of damage simulation has been shown to model field damage well [q].
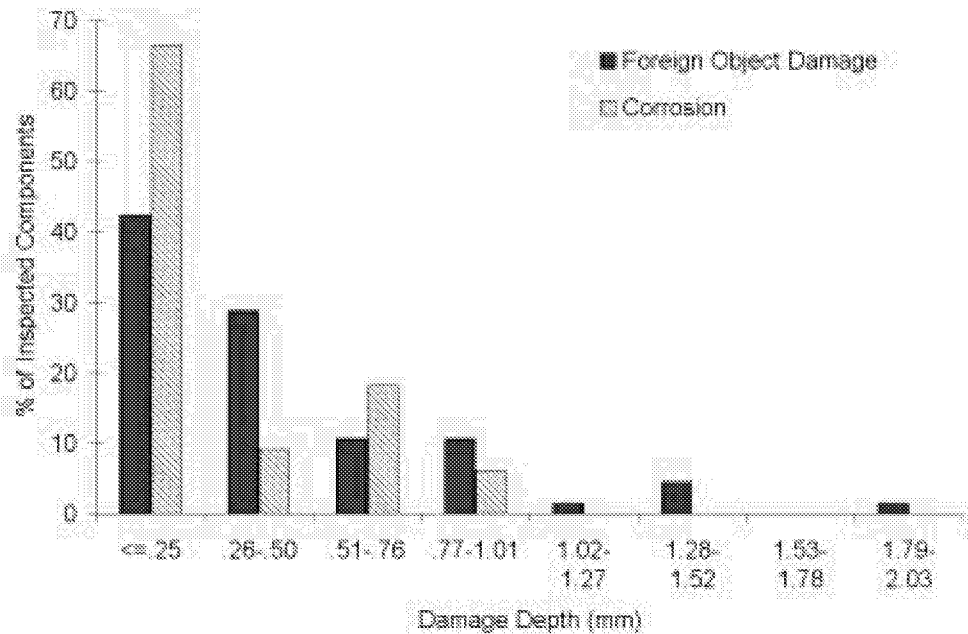
*Figure 5:* Survey of field damage for rotorcraft structure.
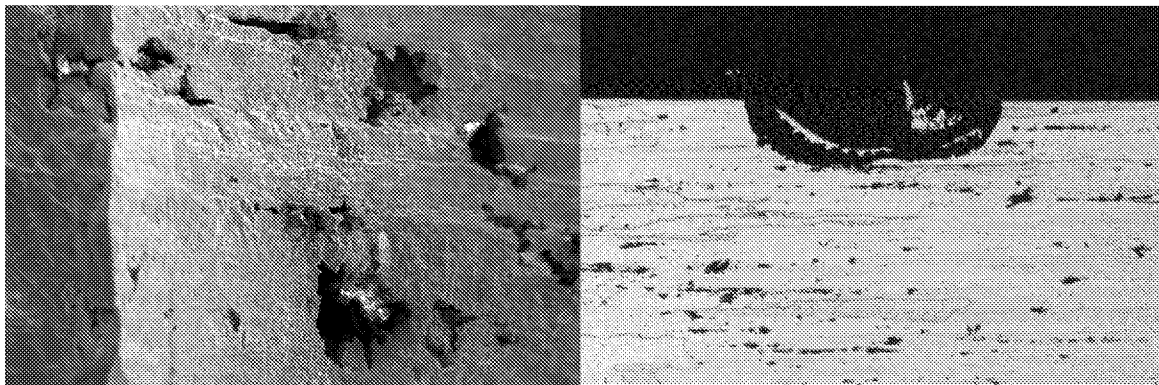
Figure 6: Photographs of field and laboratory light corrosion damage.

## Design and Life Management of a Rotorcraft Spindle Lug

An example application of the four methods of design and life management is discussed for the rotorcraft spindle lug depicted in Figure 7. The lug is required to maintain structural integrity for a minimum of 10,000 flight hours, where a single flight hour is described by the loading spectrum defined in Figure 8. The critical design dimensions of the lug are as follows: R is the outer radius of the lug and defines the attachment areas, t is the thickness of the entire structure (both the lug and cylinder), P is the applied load and r is the inner radius that is fixed to 0.1 meter

to fit the attachment pin. The lug is manufactured from an aircraft-grade aluminum alloy with a density of 2.81 g/cm³ that is used herein to estimate the structural mass of the lug from each design.
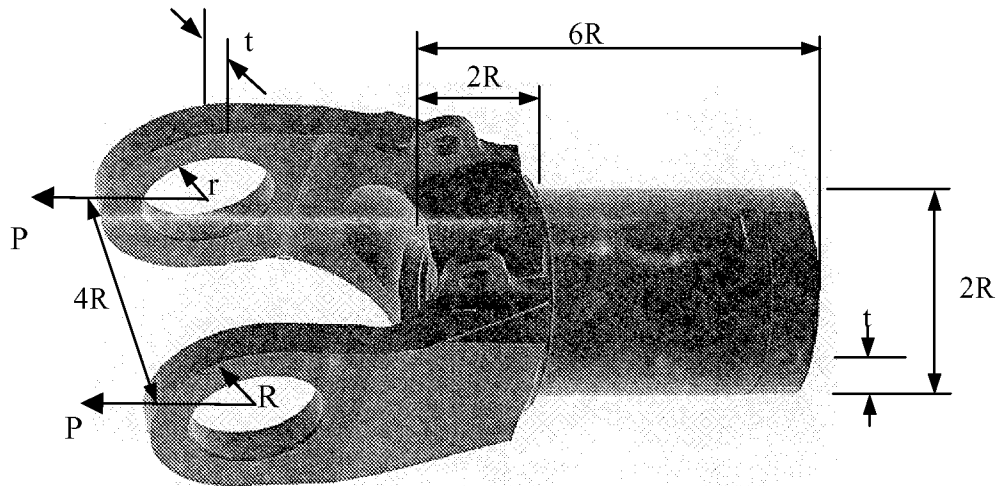
Figure 7: Rotorcraft spindle lug.
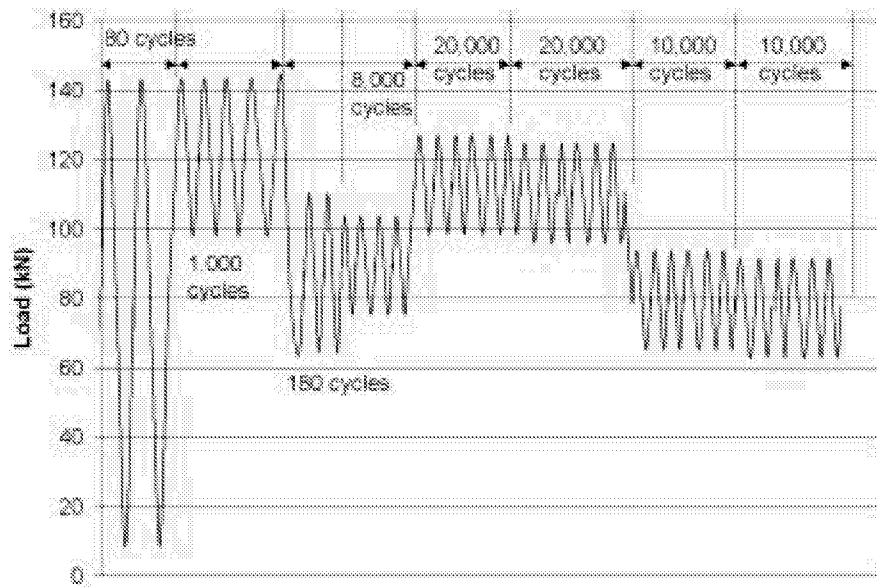
Figure 8: Loading spectrum of spindle lug used to represent one flight hour.

Stress-life Assessment

The stress-life method of design and life management is based on the linear cumulative-damage relationship described above. The margin of safety applied to the stress-life diagram has traditionally been defined by three standard deviations from the mean; that is, the mean curve that fits the data is "knocked down" by 3σ. The linear cumulative-damage relationship method

can be programmed into a spreadsheet to compute the dimensions of the lug required to meet the 10,000-hour life requirement. The stresses at the lug root were calculated from the standard solution in the literature [r]. The resulting dimensions, structural mass and maximum stress are tabulated in Table 1.

Flaw-tolerance Assessment

The flaw-tolerance method of design and life management is also based on the stress-life linear cumulative-damage relationship described above. However, the margin of safety implied in the design is based on the lower bound of the barely detectable damage data, like light corrosion and FOD. Unlike traditional stress-life analyses, a finite inspection interval is set based on the clearly detectable flaws, e.g., heavy corrosion and FOD. However, for this case the foreign-object damage is ignored for comparative purposes, based on the unpredictability of the laboratory method. The residual stresses imparted from the FOD, as shown by the improved fatigue life of the light FOD (the light FOD data has a higher number of cycles to failure than the as-manufactured data for the same stress level, as shown in Figure 1), must be understood before a component can be safely designed with this data. Further comment on the effects of using this damage is discussed later in the paper. Because the linear cumulative-damage relation is used for this method, the analyses can also be performed in a spreadsheet. The resulting dimensions, structural mass and maximum stress are also tabulated in Table 1.

Damage-tolerance Assessment

The damage-tolerance method to design and life management is defined by the time it takes an initial crack size, defined by nondestructive inspection methods, to cause the structure to fail. For the case of rotorcraft structure, the nondestructive inspection (NDI) initial crack size has been defined as 0.4 mm. The NASGRO computer code was used to predict the fatigue-crack-growth life of a 0.4 mm corner crack to failure. The crack length versus number of hours is plotted in Figure 9. The fatigue-crack-growth relationship used within the code incorporates load interaction effects of the loading spectrum [s] based on plasticity theory [t]. The material data input is a best fit of the $R = 0.7$ data depicted in Figure 2. The resulting dimensions, structural mass and maximum stress are also tabulated in Table 1.

Fault-tolerance Assessment

Fault-tolerance, much like damage-tolerance, determines the design life and inspection intervals based on fracture mechanics. However, for the case of fault-tolerance, the design of the component is based on the as-manufactured flaw size that is defined by NDI methods available to the producer. This size is typically on the order of 0.1 mm. This value is reasonable because the EIFS value computed for the as-manufactured condition was 0.1 mm at a 90% confidence level. In designing the spindle lug, the NASGRO computer code was used to predict the fatigue-crack-growth life of a 0.1 mm crack to failure. The NASGRO analysis was also repeated for the flawed conditions and is plotted in Figure 10. The light foreign-object damage and corrosion generated EIFS values that were comparable to the as-manufactured condition, however, the probability of detecting these forms of damage over cracks is significantly higher than a 0.1 mm crack, as shown in Figure 11. However, the probability of detection (POD) for this damage is still not to the level that an operator would feel safe relying on this technology. Therefore, inspecting for heavy corrosion will provide a safer operating environment and is denoted as HD, heavy damage, along with the light damage, LD, in Table 1. The resulting dimensions, structural

mass and maximum stress are tabulated in Table 1. The heavy and light FOD damage is once again ignored, as with the flaw-tolerance method, for comparative purposes but will be commented on in the discussion section.
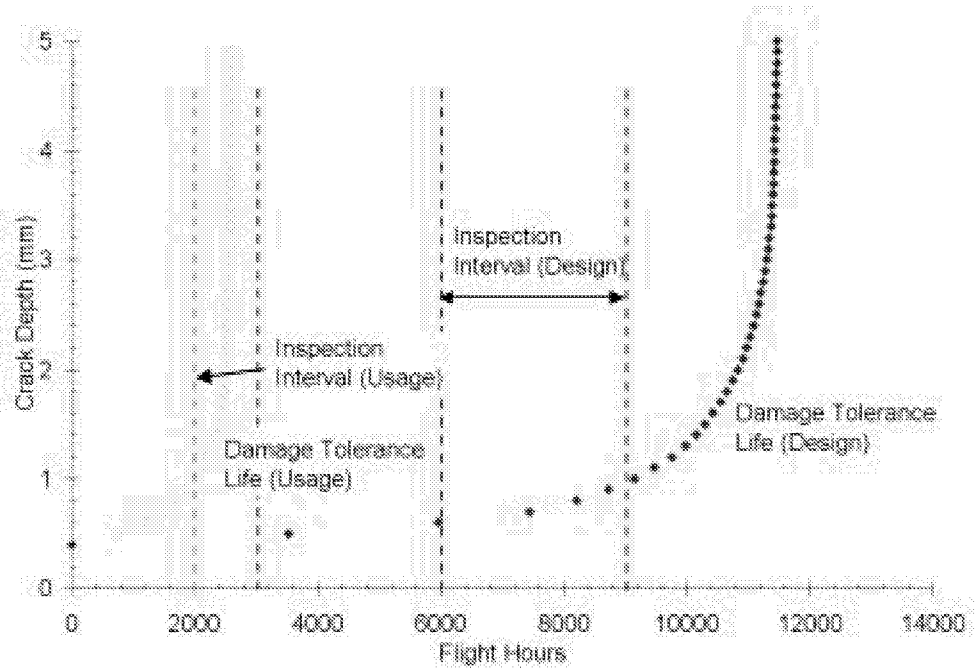
Figure 9: Fatigue crack growth time to failure for the damage-tolerance method.
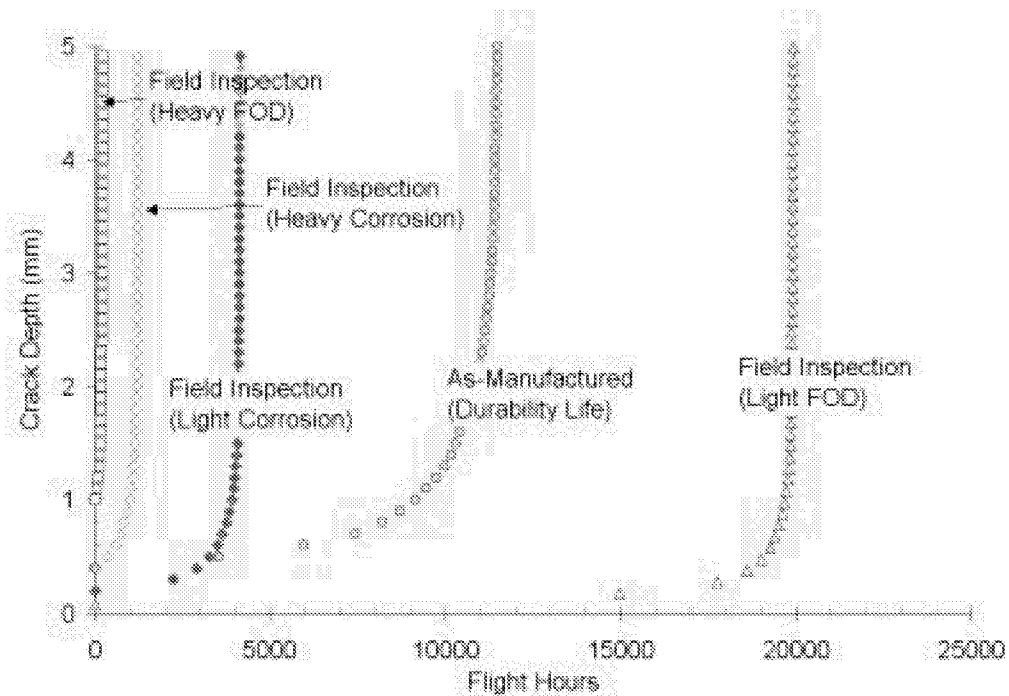
Figure 10: Fatigue crack growth time to failure for the fault-tolerance method.
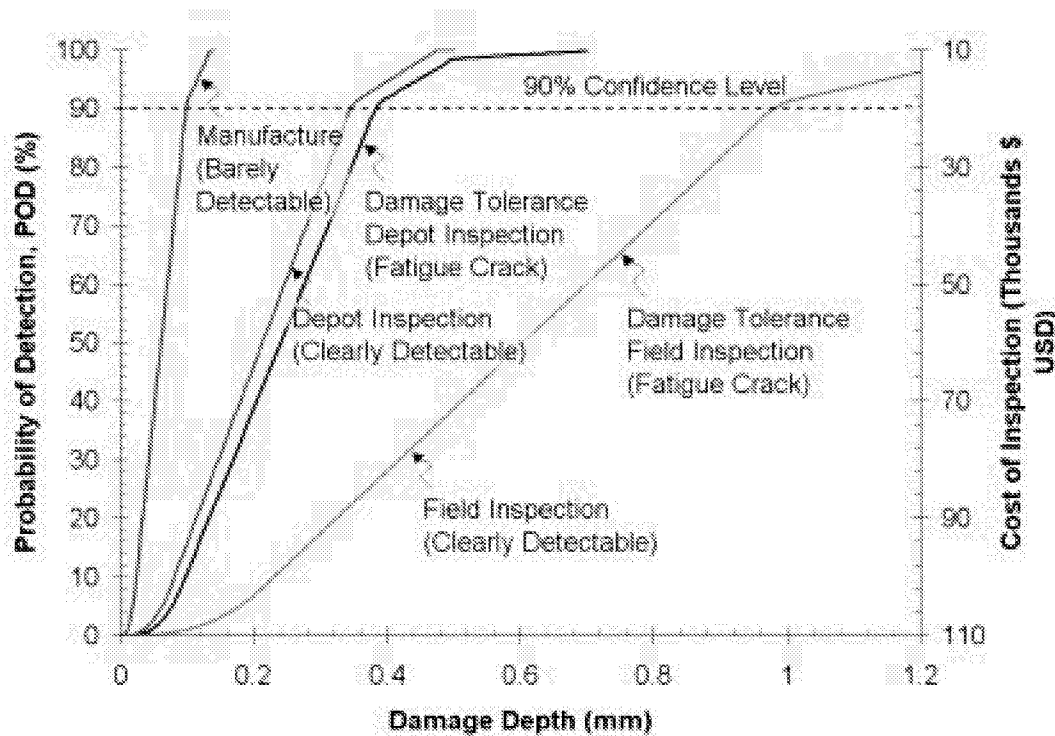
Figure 11: Probability of detection for various damage states and relative cost.

*Table 1:* Spindle lug dimensions, mass, stress and inspection interval (if required)

	Safe-life	Flaw-tolerance	Damage-tolerance	Fault-tolerance
Outer Radius (m)	0.250	0.250	0.250	0.250
Thickness (m)	0.021	0.016	0.067	0.044
Mass (kg)	100.0	97.2	123.2	112.1
Max. Stress (MPa)	34.53	45.31	10.35	16.48
Inspection time (hours)	Not Applicable	6,000	3,000	1,000 HD 4,000 LD

Spindle Lug Economics

The implementation of new technologies into design and life management are driven by economics and safety. Therefore, the implementation of the flaw- or fault-tolerance method must be weighed versus traditional safe-life and damage-tolerance methods. Furthermore, the economics of component inspection and replacement must be weighed by both the manufacturer and operator. For the case of a spindle lug, it is assumed that the manufacture and delivery of a single lug to an operator is 25,000 dollars plus 1,000 dollars per kilogram (all estimates contained herein are in United States Dollars). The cost of an operator to inspect the component is a function of the NDI method applied, whether NDI is performed at depot or in the field, and the size of the damage being inspected for. A summary of the economics of NDI is plotted in Figure 11 for the case of the spindle lug.

Investigating the realities of implementing new design and management technologies requires some general assumptions that are based on the actual usage of rotorcraft:

1. The component will be operated for three times the original design life.
2. The actual flights loads are thirty percent higher (vibratory stress) than the design spectrum.
3. The NDI in the field is not being performed in the field at the defined intervals. However, the NDI at depot, if prescribed, is regular.
4. The lug will be made available at depot for a comprehensive inspection every 2,000 hours, corresponding to the estimated component-life expectancy. The risk of incurring damage between depot intervals is plotted in Figure 12.
5. Life extension methods have been implemented. All light corrosion and FOD will be removed at depot and is assumed to behave like the as-machined structure for the purpose of this study.
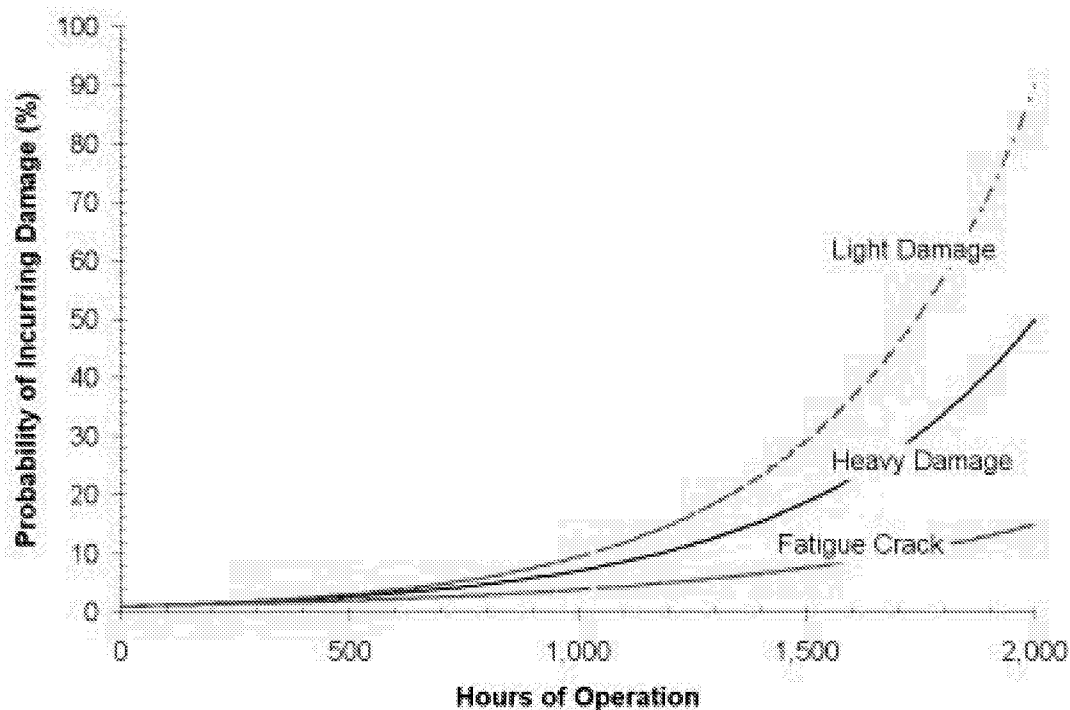

Figure 12: Risk of incurring damage between depot intervals.

Based on these assumptions, the economics of the spindle lug can alter drastically. The traditional stress-life method is dependent upon only the new lifetime and stress requirements. Increasing the stress levels by thirty percent within the lug will decrease the usable life from 10,000 hours to 4,060. Therefore, the component will need to be replaced 7 times, as the component was originally designed to be replaceable at depot every 10,000 hours, but will now be replaced every 4,000 hours at depot. Because there is no inspection mandated in a stress-life program, the lack of inspection has no bearing on the affordability of the lug. However, as the risk of damage increases, the component will most likely fail 71% of the time from damage prior to depot. The scheduled replacement of the lug will increase the cost of operation by 700%. The unscheduled replacement could add an additional 350% to the original cost, lawsuits and loss of the helicopter not withstanding. This information is recorded in Table 2.

The damage-tolerance analysis based on the 0.4 mm crack is affected by the usage increase and depot schedule. Timing the inspections with depot could save considerable funds. The increased stress levels changed the inspection interval from 3,000 hours to 650 hours, as shown in Figure 9. Based on the data presented in Figure 10, the cost of inspection in the field is $80,000, whereas the original design could be performed at depot with a cost of $20,000. However, the field inspections will not be performed, but inspection will be done at the depot. The lack of inspection may save costs, but significantly increases the risk of a field failure. Fortunately, the damage-tolerance design life is greater than the inspection interval, allowing one inspection prior to the predicted failure time. A crude risk assessment could be based on the probability of incurring damage between depot inspections, as depicted in Figure 13. Based on the increased risk of failure, the risk assessment would predict that the probability of a field failure would be 50%, since only one-third of the inspections are being performed. The true cost of operation is summarized in Table 2.

The flaw-tolerance method is affected by all of the assumptions. The stress-life method yields a decrease in overall retirement time and inspection interval, based on the increase in vibratory stress. Increasing the stress levels within the lug will decrease the usable life from 10,000 hours to 4,400. However, the removal of the light corrosion and FOD every 2,000 hours, if detected, is a considerable cost savings as this damage no longer governs retirement, returning the retirement time to 10,000 hours. Therefore, the component will need to be replaced twice since the component was originally designed to be replaceable at depot every 10,000 hours. The inspection intervals defined by heavy corrosion were originally every 6,000 hours; with the increased usage the inspection interval is 800 hours. Based on the risk of damage presented in Figure 12, the probability of a field failure is 93%, as the inspections are being omitted and the part was designed with these in mind. Inspecting the component every 2,000 hours instead of 6,000 will increase the cost of operation by 300%, which is recorded in Table 2.

Finally, the fault-tolerance method is impacted by the increase in loads and the lack of inspection. The increased stresses changed the inspection interval for light and heavy damage from 4,000 / 1,000 hours to 900 / 300 hours, as shown in Figure 13. However, these inspections will not be performed in the field. Analogous to the damage-tolerance method, the inspection for damage must be coordinated with the depot. For this case, the blending of the light damage from the lug leaves a component that is nearly in the as-manufactured condition, effectively resetting the overall durability life. Based on the probability of incurring damage between depot services, the risk of premature failure is 60%. A summary of the economics of the fault-tolerance method is given in Table 2.
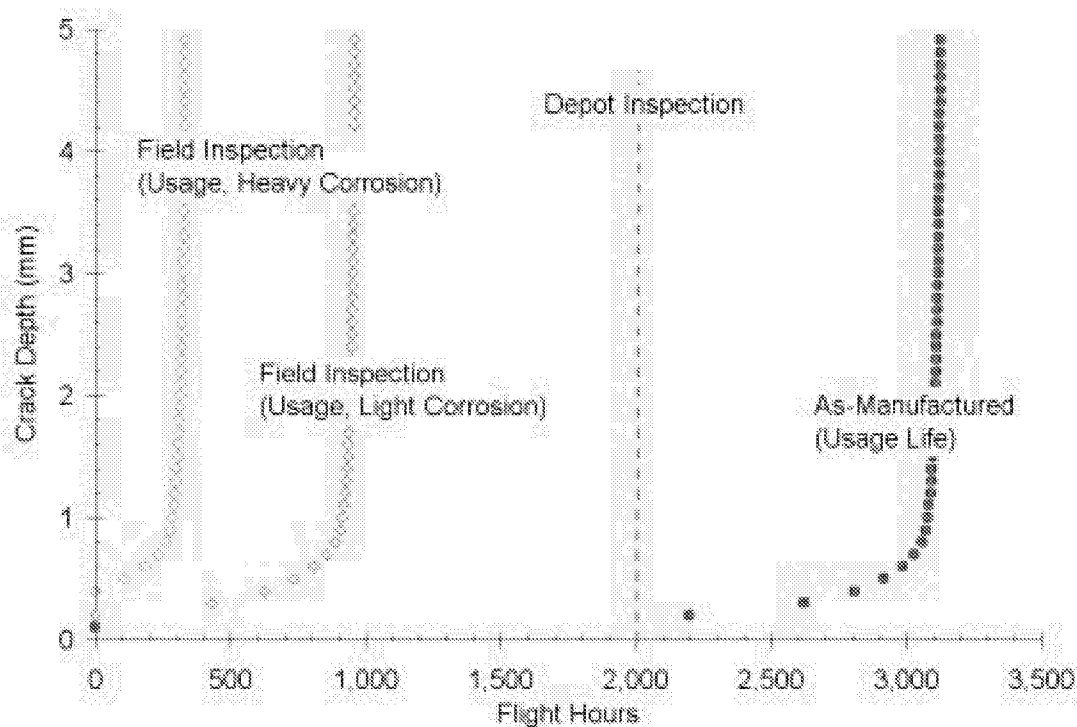
Figure 13: Life management for the fault-tolerance method.

Table 2: Spindle Lug economics of ownership (Figures in $1,000's).

	Safe-life	Flaw-tolerance	Damage-tolerance	Fault-tolerance
Original design cost	125	122	148	137
Original inspection cost	Not Applicable	60	240	240
New design cost	875	172	148	137
New inspection cost	Not Applicable	300	300	300
Probability of Failure	71%	93%	50%	60%

Discussion

Changes in usage and operating environment are commonplace in civil and military applications. The design of rotorcraft flight-critical components must be robust enough to encompass realistic usage spectra that the aircraft is not designed for. For the spindle lug example presented herein, the traditional stress-life method to design and life management performed poorly. The component would likely fail several times in the field, incurring loss of property and life. The flaw-tolerant stress-life method only increased the risk of a catastrophe by designing a part based on inspections that must be performed to maintain safety. The design initially reduces the cost to the operator through lower weight, but any variability in usage and environment will completely degrade safety. The fracture-mechanics-based damage-tolerance

method adapted to the altered usage environment well. However, the lack of inspection increased the risk of failure by 200%. The fault-tolerance method was also hindered by inspection. The risk of failure increased by 250%, more than the comparable damage-tolerant component, because the initial design was 10% lighter and designed for more frequent inspection.

The impact of foreign-object damage would show all of these design methods as unreasonable. If heavy FOD behaved the same as the laboratory specimens (Fig. 1), all of these designs would fail in less than 500 hours. In service, foreign-object damage imparts residual stress fields, displaces material and can initiate fatigue cracks instantaneously. It is unrealistic to attempt to define FOD in a laboratory setting when the research community cannot agree upon the effect of shot peening, which is a controlled process. Therefore, the FOD damage presented herein is useful for reference and simple what-if scenarios, however it is useless for design. It is important for a company that intends to incorporate flaw- and fault-tolerance methods to keep this fact in mind.

The rotorcraft manufacturer has the challenge of producing a safe and cost effective aircraft. Recent research into alternative design methods is a testament to the continued drive to improve future products. The traditional stress-life method has proven that the empirical method is conservative and safe, since there are few accidents. However, this method does not account for any variation in usage, and if the part is damaged in service it can readily fail since no inspection plan is required. Based on the spindle-lug example presented herein, a reasonable risk of damage is assumed that shows the shortcomings of the stress-life method. The cost of operation is increased 7 times, and the risk of failure is increased by 3 ½ times.

The development of flaw-tolerance methods as a replacement for stress-life methods appears reasonable. It is logical that better quantification of usage during the design will lead to a more robust structure and inspection will provide a safer operating environment. However, it is difficult to truly quantify the usage environment for any single component, and to do such for every part on a rotorcraft will be very time consuming and expensive. Further, the flaw-tolerance method does not reduce the level of empiricism that is a primary reason for abandoning the stress-life method. The flaw-tolerance method to design and life management, as shown by the spindle lug example, may not be ideal for rotorcraft design because changes in usage can impact safety tremendously. For this case, the probability of the component failing in the field is 93%.

Implementation of damage-tolerance methods into the high-cycle fatigue arena has been a significant challenge. The aircraft engine manufacturers have been developing methods and researching implementation methods for over a decade. The rotorcraft industry will also implement damage-tolerance successfully. It will simply take time for the nondestructive inspection methods to become practical and cost-effective. However, as shown by the spindle lug example, the damage-tolerance method is not overly severe when the economics of operation are considered. Furthermore, changes in usage can be accommodated with little additional cost and, if inspections are performed, little impact on safety.

Finally, alternative methods, such as fault-tolerance methods, are meant to improve the applicability of damage-tolerance concepts to high-cycle fatigue systems by lessening the deficiencies in inspection methods. The basis of the method is damage tolerance, which will provide a better estimate for durability life and "retirement for cause" than the stress-life methods. However, the method suffers some of the same shortcomings of flaw-tolerance methods. It will be very difficult and expensive to quantify the usage environment and

subsequently the probability of damage of an aircraft that has yet to fly. However, with advances made in probabilistics, risk management and experimental methods, a conservative method can be developed. As NDI methods become available to detect cracks of 0.1 mm or less in the field, it would not be unreasonable to transition from fault- to damage-tolerance methods. However, it has been shown throughout aviation history that as the NDI methods find smaller flaws, there will be a design that requires finer resolution. It is in these new designs that a tool like a fault-tolerance method may be fitting.

Conclusions

The transition from stress-life management of rotorcraft to that based on damage-tolerance will not by easy. There is significant history clearly illustrating stress-life methods as safe and reliable. However, rotorcraft manufacturers must still rely upon damage-tolerance methods to explain service failures and further maintain the safety of the fleet. It is logical that the adoption of damage-tolerance principals into the design will save the company development funds, and additional costs when a component unexpectedly fails. Furthermore, as illustrated by the spindle lug example presented herein, the cost of operating a damage-tolerance designed part is significantly less in the long term, even with changes in the usage environment. Furthermore, the continued zeal for stress-life methods, such as flaw-tolerance methods, will cost operators and manufacturers both profit and safety. This fact will be replayed every time a manufacturer is required to develop damage-tolerance data to manage a field issue, and an operator is hindered by unmanageable inspections. Implementation of damage-tolerance design and management methods in rotorcraft is possible. Rotorcraft companies prepared to adopt damage-tolerance methods will manufacture better, safer products and increase profits for themselves and their operators.

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